# ASTEROID RECONNAISANCE AND SAMPLE RETURN USING SOLAR ELECTRIC PROPULSION

Aron A. Wolf,
Supervisor, Inner Planets Mission Analysis Group
MS 301-140H
Jet Propulsion Laboratory
Pasadena CA 91109

Reconnaisance of asteroids has thus far been accomplished on a limited scale. The scientific community has expressed interest in conducting long-term reconnaisance of multiple mainbelt asteroids and returning a sample from a mainbelt asteroid, with Ceres and Vesta the targets of greatest interest. This work presents results of preliminary feasibility studies of these missions. Since propellant requirements are prohibitively large with chemical propulsion, Solar Electric Propulsion (SEP) was assumed here. Multiple opportunities for Ceres/Vesta reconnaisance missions and Vesta sample returns were found which appear to be feasible in the next decade.

## INTRODUCTION

Study of multiple mainbelt asteroid rendezvous and mainbelt asteroid sample return missions was undertaken to determine feasibility using a launch vehicle in the Delta 7925 class, and to assess technology development requirements for these missions. Efforts concentrated on missions to Vesta, Ceres, and Psyche, which were taken to be the targets of greatest scientific interest. A representative set of scientific objectives was defined for each mission. Launches between 2003 and 2008 were explored. A 2008 launch allows technology development until a cutoff date of roughly 2005.

Preliminary estimates of spacecraft mass were made to assess whether the spacecraft could fit comfortably within the launch vehicle requirement. As is appropriate for early mission studies like this one, these estimates relied heavily on information developed for previous missions and studies, with some conservatism applied in place of analysis that would be part of any more extensive effort.

Several cost-saving measures that have become typical of Discovery-class missions were assumed. Personnel from spacecraft design, development, integration and test become part of the operations team. Command and telemetry software developed for operations is also used in ATLO. During interplanetary cruise, no science observations requiring science team support are made, and DSN tracking is minimized (one 8-hr. pass per week to support routine operations, with continuous tracking as needed after launch and for special events).

## MULTIPLE MAINBELT ASTEROID TOUR

A SEP rendezvous with Vesta and Ceres appears feasible with current technology for a launch in 2005. Flight time to Vesta is about 2 years, followed by a 300-day stay at Vesta, departure to Ceres, and a 300-day stay at Ceres. Total mission duration, including the 300-day stays at Vesta and Ceres, is about 7 years.

The assumed scientific objectives for this mission were **encounters with more than one asteroid**; **global mapping** at visible wavelengths at resolutions of 10m-100m, and in the near-IR at 100m-1km resolution; **composition** (via X-ray and/or gamma-ray measurements of elemental abundances); and **gravity, mass and density** via radio science.

Two mission modes were examined. In the first (called **rendezvous and orbit**), all scientific observations (visible/near-IR imaging, global gamma ray and/or X-ray observations, and radio science) are made from orbit. This is the most desirable option because it allows global determination of elemental composition. About 300 days in orbit about each asteroid is needed to accumulate enough data to arrive at reasonably accurate estimates of abundances of several elements.

In the second option (called **rendezvous and orbit with surface packages**), only imaging and radio science are conducted from orbit. A surface package (or packages) equipped with gamma-ray and/or X-ray instruments lands and makes local observations of elemental abundances which are relayed to Earth through the orbiting spacecraft. Consequently, stay time at each asteroid can be shorter, reducing mission duration to about 5 years. This returns less science than the previous option because observations of elemental composition are made locally rather than globally. Increased operations costs of the orbital-only mission must be traded against the cost of building the surface package(s).

Instruments

**SEP rendezvous and orbital observations.** The instrument payload consists of a visible imager/IR spectrometer based on the PICS instrument design (8 kg) and an X-ray/Gamma ray spectrometer identical to the one flown on the NEAR (Near Earth Asteroid Rendezvous) mission (26.9 kg), for a total mass of 34.9 kg.

Both the X-ray and gamma ray spectrometers need a significant amount of time in orbit about the asteroid to accumulate enough data to arrive at reasonably accurate estimates of abundances of several elements. For instruments of equivalent capability to NEAR, abundances of perhaps six elements can be measured after a year in orbit at an altitude of 10 asteroid radii or less. Spatial resolution of GRS data after a year in orbit at that altitude is roughly estimated at one-fifth of the spacecraft's altitude, and spatial resolution of X-ray data is estimated at 1/10 - 1/15 of the spacecraft's altitude.

**SEP rendezvous and orbit with surface package(s).** A previous comet mission study estimated the mass of a GRS for surface observations on a comet at 2 kg; mass of the X-ray spectrometer required for surface observations is estimated at about 3 kg, making the total payload mass for the surface package 5kg.

The surface package must have a chemical propulsion system with enough propellant to deorbit, and then to slow itself prior to impact. Assuming an orbital altitude of 100km at Vesta, the total $\Delta V$ requirement for the surface package is about 400 m/s.

For comparison, penetrator designs suggested for previous comet mission studies have been estimated to weigh about 45 kg; the Rosetta lander's mass is expected to be in the 65 kg range, and the ST4/Champollion lander's mass was over 100 kg; however, these all are more complex devices with more instruments and/or sample acquisition systems not required here. A guesstimate of 30 kg was adopted as the dry mass estimate for a surface package. To this was added 3.7 kg of propellant (Isp=300) to provide 400 m/s $\Delta V$, bringing the total "wet" mass of the surface package to 33.7 kg.

## Mission design

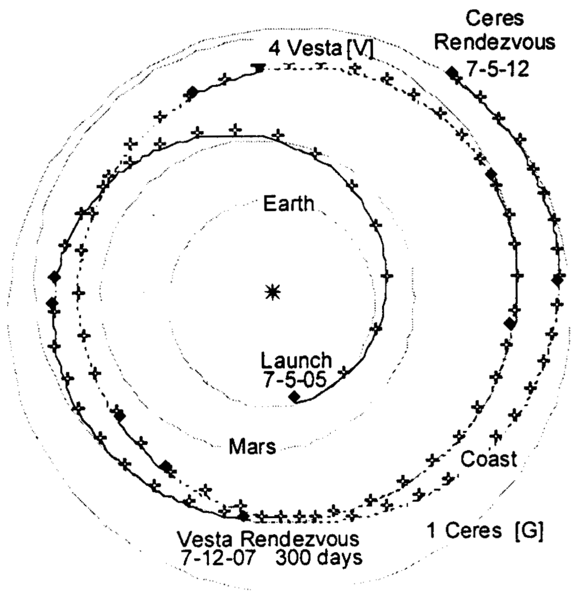
Table 4 shows some parameters of interest for three Vesta-Ceres rendezvous opportunities in the time frame of interest. These trajectories are illustrated in Figs. 1 through 3.

*Table 4: Vesta-Ceres rendezvous trajectory data*

| First target | Second target | Launch date | Arr date at first target | Arr date at second target | Flight time to first target (yr.) | Stay time at first target (d) | Flight time to 2d target (yr.) | Stay time at 2d target (d) | Total mission duration (yr.) | Launch C3 (km2/s2) | Launch mass (wet) | Xenon mass (kg) | S/C dry mass at launch (kg) |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 4 Vesta (V) | 1 Ceres (G) | 7/5/05 | 7/12/07 | 7/5/12 | 2.02 | 300 | 7.00 | 300 | 7.82 | 2.02 | 1138 | 434 | 704 |
| 4 Vesta (V) | 1 Ceres (G) | 12/3/06 | 10/25/08 | 12/30/12 | 1.89 | 200 | 6.08 | 200 | 6.62 | 4.02 | 1090 | 398 | 692 |
| 4 Vesta (V) | 1 Ceres (G) | 3/14/08 | 2/23/11 | 1/1/14 | 2.95 | 60 | 5.80 | 60 | 5.97 | 1.10 | 1060 | 407 | 653 |

These trajectories were generated under the following assumptions:

- 10kw (end-of-life at 1AU) SEP system
- 10% launch vehicle margin
- Use of NSTAR thrusters of 30-cm diameter (identical to those used on DS1)
- 2 or 4 thrusters firing simultaneously during thrust arcs
- High-efficiency Silicon solar array
- 95% SEP "duty cycle" (i.e. thrusters assumed firing at only 95% of their design thrust level to account for shutdown periods due to operational necessity)
- 100W electrical power delivered to the spacecraft (exclusive of the SEP system)




300 day stay time at Vesta
50 kg mass dropped at Vesta
2 or 4 thrusters operating simultaneously

| | |
|---|---|
| Launch vehicle | Delta 7925 |
| Launch energy (km2/s2) | 2.0 |
| Dec. launch asympt. (deg.) | -13.7 |
| Flight time (yrs) | 7.0 |
| Total chemical ΔV (m/s) | 200.0 |
| cruise (incl. nav) | 50.0 |
| Mnvrg @ asteroids | 150.0 |
| | |
| LAUNCH VEH. CAPABILITY (KG) | 1251.8 |
| LV margin (%) | 10.0 |
| LV margin (kg) | 113.8 |
| ALLOWED INJECTED MASS (KG) | 1138.0 |
| Total propellant | 506.5 |
| SEP | 434.0 |
| Chemical | 72.5 |
| ALLOWED DRY S/C MASS (KG) (assuming 10% launch vehicle margin) | 631.5 |

30 day tics on spacecraft

*Fig. X: 2005 7-year Vesta-Ceres Multiple Mainbelt Asteroid Rendezvous*

The stay time at Vesta (which must be long to complete GRS and X-ray measurements from orbit) is one of the factors determining the amount of Xenon needed for the mission, since it determines the departure time from Vesta and the shape of the

Vesta-Ceres trajectory segment. However, the stay time at Ceres does not affect the Xenon propellant requirement, because Ceres is the final destination. The length of time at Ceres we choose to include in the mission is subject only to science data-gathering and perhaps cost considerations. Here, the stay time at Ceres was arbitrarily assumed to be equal to the stay time at Vesta.

The trajectory launching in 7/05 with a 300-day stay time at Vesta (the first line in Table 4) is discussed in detail here.

Spacecraft design

A chemical propulsion system is required in addition to the SEP system, to provide attitude control when SEP thrusters are off and to maneuver in the vicinity of the asteroids. The table in Fig. X shows that assuming 200 m/s chemical $\Delta V$ is required, the maximum allowed dry mass of the spacecraft at launch is 631.5 kg on the 2005 trajectory.

Mass summaries are presented in Table 6 for the cases with and without surface packages. Keeping a 10% launch vehicle margin allows only a 13% mass growth contingency without a surface package, and a 11.2% contingency with a surface package. These contingencies are small. However, a 10% launch vehicle margin is overly conservative for the venerable Delta 7925, and the spacecraft carries no new technology. Also, at this early stage of analysis conservatism is applied in lieu of design detail in arriving at mass estimates for several subsystems. Finally, no effort was made to find the SEP system power level that maximizes mass performance for any of these trajectories. Examining power levels other than 10kw should improve performance.

| | Rendezvous / Orbital Observations | | | With surface package | | |
|---|---|---|---|---|---|---|
| | Mass (kg) | % of dry mass (no cont.) | % of launch mass (no cont.) | Mass (kg) | % of dry mass (no cont.) | % of launch mass (no cont.) |
| Instruments | 34.0 | 6.1% | 3.0% | 8.0 | 1.4% | 0.7% |
| Surface package | 0.0 | 0.0% | 0.0% | 33.7 | 5.9% | 3.0% |
| **Payload total** | **34.0** | **6.1%** | **3.0%** | **41.7** | **7.3%** | **3.7%** |
| | | | | | | |
| Attitude Determination & Control | 16.0 | 2.9% | 1.4% | 16.0 | 2.8% | 1.4% |
| Command & Data Handling | 10.9 | 2.0% | 1.0% | 10.9 | 1.9% | 1.0% |
| Power | 121.3 | 21.7% | 10.7% | 121.3 | 21.4% | 10.7% |
| Propulsion-Chemical | 26.4 | 4.7% | 2.3% | 26.4 | 4.6% | 2.3% |
| Propulsion-SEP | 158.8 | 28.4% | 14.0% | 158.8 | 28.0% | 14.0% |
| Structure | 133.5 | 23.9% | 11.7% | 134.9 | 23.8% | 11.9% |
| Cabling | 29.0 | 8.0% | 2.5% | 29.0 | 8.0% | 2.5% |
| Telecom | 21.6 | 3.9% | 1.9% | 21.6 | 3.8% | 1.9% |
| Thermal | 7.4 | 1.3% | 0.7% | 7.4 | 1.3% | 0.7% |
| **Bus Total** | **524.9** | **93.9%** | **46.1%** | **526.3** | **92.7%** | **46.2%** |
| | | | | | | |
| **Spacecraft Total (Dry)** | **558.9** | **100.0%** | **49.1%** | **568.0** | **100.0%** | **49.9%** |
| Contingency | 72.6 | **13.0%** | 6.4% | 63.7 | **11.2%** | 5.6% |
| **Spacecraft with Contingency** | **631.5** | **113.0%** | **55.5%** | **631.7** | **111.2%** | **55.5%** |
| Chem. Propellant & Pressurant | 72.5 | 13.0% | 6.4% | 72.5 | 12.8% | 6.4% |
| SEP Propellant | 434.0 | 77.6% | 38.1% | 434.0 | 76.4% | 38.1% |
| **Launch Mass** | **1138.0** | **203.6%** | **100.0%** | **1138.2** | **200.4%** | **100.0%** |
| | | | | | | |
| **Launch Vehicle Capability** | **1251.8** | **224.0%** | **110.0%** | **1251.8** | **220.4%** | **110.0%** |
| **Launch Vehicle Margin** | **113.8** | **20.4%** | **10.0%** | **113.6** | **20.0%** | **10.0%** |

*Table X: Spacecraft mass summary, Vesta-Ceres rendezvous with and without surface package*

Three-axis stabilization is preferred over spin stabilization for this mission due to the presence of the large solar panels required for SEP. The attitude determination and control subsystem includes redundant star cameras, sun sensors, and IMUs, all with flight heritage. The C&DH subsystem uses the R6000 processor. Solar panels provide 10kw of power (end of life at 1AU) at an assumed 100 watts/kg. Battery requirements (5 amp hr. at 2.9 amp-hr/kg) and power electronics necessitate no new technology developments.

SEP hardware identical to that used on the DS1 mission was assumed. The trajectory was computed assuming either 2 or 4 thrusters are operating simultaneously during thrust periods; therefore, no less than 4 thrusters can be carried. Thruster lifetime, expressed in Xenon throughput, is a concern for SEP missions. To complete this mission, 434kg of Xenon is required, which equates to a throughput of 86.8kg of Xenon per thruster if 5 thrusters are carried. This is conservative, given that the DS1 SEP hardware was ground-tested to 83kg throughput before the launch of DS1, and that testing is presently in progress to qualify those thrusters to 120kg throughput.

Structural mass (including cabling) was estimated to be 18% of non-structural injected ("wet") mass. Cabling was estimated at 8% of non-structural dry mass. The telecom subsystem incorporates a high-gain dish (with a two-axis gimbal to enable Earth pointing), a medium-gain horn, and two low-gain patch antennas.

Cost

Cost estimates from previous Team X studies and comparisons of factors affecting cost between this study and those previous studies indicate that the cost of this mission should fall near the top of the range of costs associated with Discovery missions.

## MAINBELT ASTEROID SAMPLE RETURN

The principal scientific objective assumed for this mission was **sample return** from multiple locations within a few meters of each other, returning 10g of material at each collection site with 0.5-cm pieces or larger. Meteorites can have grain structures occasionally as large as 0.3cm; samples obtained from the asteroid must try to preserve this structure so it can be studied. Preservation of core stratigraphy is desired, with core samples taken from 1m depth or less. Preservation of the sample at a maximum temperature of 50 deg. C and vacuum at $10^{-3}$ torr or less until an hour after landing on Earth is also required to preserve hydrated materials.

**Global mapping** at visible and near-IR wavelengths, **global composition** with X-ray and/or gamma-ray observations, and **gravity, mass and density** via radio science were assumed to be secondary objectives. These observations are taken from orbit as discussed for the multiple mainbelt asteroid rendezvous mission.

To collect sufficient GRS/X-ray data to determine global composition, the spacecraft must remain in orbit for about a year after imaging is complete and the sample is collected. The primary objective of returning a sample is achievable with a shorter stay at the asteroid. Shortening the stay time could be considered as a descope option.

Several sample retrieval methods are possible. The method used here involves a lander which detaches itself from the orbiting spacecraft and descends and lands autonomously anchoring itself securely to the surface. After the sample is collected, a portion of the lander containing the sample returns to the spacecraft. After docking, the sample is placed in the sample return capsule.

An alternate method involves landing the entire spacecraft on the surface of the asteroid, eliminating the need to build a separate lander. Landing the spacecraft with its large solar panels deployed is a complex undertaking which requires careful consideration of the dynamics involved as well as a larger safe landing zone on the asteroid.

Another method employs small penetrators (~8 kg each), each of which collects a sample upon impact. The forward portion of the penetrator buries itself in the asteroid; the rear portion (containing the sample canister) ejects itself from the surface. Sample canisters are retrieved via rendezvous (with the orbiter chasing the canisters) or via a tether. Canisters are ejected from the asteroid's surface one at a time, with sufficient time (several days, in the rendezvous case) provided to reposition the orbiter to prepare for the next ejection. Both rendezvous and tethered returns of small penetrators have been examined in previous comet lander studies.

Instruments

The orbiter's instrument payload was assumed identical to that discussed for the multiple mainbelt asteroid mission. The lander carries the ISIS and CIRCLE imagers developed for the ST4/Champollion comet lander, and sample collection equipment. No in-situ sample analysis is performed on the asteroid's surface.

Several methods of sample collection, all of which require significant development, have been discussed in previous studies of both asteroid and comet sample return missions. These are:

- A drill (~8 kg) similar to the one intended for use on the ST4 comet lander, returning samples from ≤1m. This is the furthest along in development.
- An ultrasonic drill, currently in early stages of technology development at JPL.
- A "deep" (10m) drill, currently in early stages of technology development at JPL.
- Chippers and corers. Chippers use two counter-rotated circular saws on the end of a robotic arm which dig into surface rock and spray chips of material into collection baskets. The baskets trap the chips with honeycomb and/or aerogel. Corers are fixed cylindrical containers on the end of rigid arms, fired downward by an explosive charge. Corers have been shown to work in cement, and should easily collect sand as well.
- A subsurface explorer which burrows below the surface. A prototype has been successfully tested in sand.

For the purposes of arriving at a preliminary estimate of mission feasibility, 20 kg was allocated for sample collection hardware to be carried on the lander, with the implicit assumption that one of the schemes discussed above could be implemented within that allocation. Changes in science requirements related to sample collection (e.g. sample size, number of samples, depth from which they must be collected) could have a great deal of influence on the mass of this hardware.

## Mission design

Figs. 1 through 3 illustrate several Vesta sample return opportunities launching in the time frame of interest. Some pertinent data on these trajectories appears in Table 4.

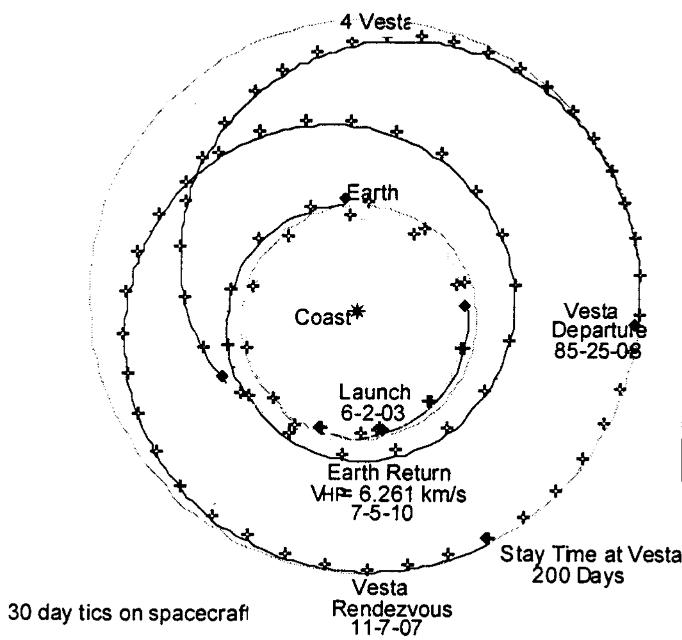
*Table 4: Vesta sample return trajectory data*

| Target Body | Launch date | Arr date at target body | Earth return date | Flight time to target (yr.) | Stay time (d) | Total flight time (to Earth return) (yr.) | Launch C3 (km2/s2) | Launch mass (wet) | Xenon mass (kg) | S/C dry mass at launch (kg) | Entry velocity at Earth return (km/s) |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 4 Vesta (V) | 6/2/03 | 11/7/07 | 7/5/10 | 4.43 | 200 | 7.09 | 3.25 | 1108 | 510 | 598 | 12.8 |
| 4 Vesta (V) | 5/26/04 | 7/27/07 | 6/30/10 | 3.17 | 360 | 6.09 | 19.73 | 788 | 303 | 485 | 13.2 |
| 4 Vesta (V) | 7/3/05 | 8/22/07 | 7/5/10 | 2.14 | 360 | 5.00 | 20.81 | 771 | 285 | 486 | 13.2 |

The assumptions used to generate these trajectories were the same as those used for the multiple mainbelt asteroid rendezvous trajectories except that a 6.75kw SEP system was used instead of a 10kw system, 250W electrical power to the spacecraft was assumed instead of 100W, and a 90% SEP "duty cycle" was used instead of 95%.

It is apparent that, as often occurs in mission design, flight time trades against mission performance. The opportunities with shorter flight times (2004 & 2005) require much higher launch energies than the one with the longest flight time (2003), which contains one more “loop” around the Sun than the 2005 trajectory. The increase in launch energy is more than enough to cancel out the advantage of almost 200kg in the Xenon propellant requirement held by the shorter trajectories over the 2003 opportunity.

The 2003 trajectory (the first line in Table X) is discussed in detail here. This trajectory offers significantly better mass performance than the other two. Although 2003 may be too soon to launch this mission, it is anticipated that other opportunities roughly resembling the 2003 opportunity (i.e. launch energy in the 3-5 $km^2/s^2$ range and flight time of approximately 7 years) exist in later years. (The list of trajectories in Table X is not a complete list of all trajectories available between 2003 and 2008.)




200 day stay time at Vesta
50 kg mass dropped at Vesta
2 or 4 thrusters operating simultaneously

| | |
|---|---|
| **Launch vehicle** | **Delta 7925** |
| **Launch energy (km2/s2)** | **3.3** |
| **Dec. launch asympt. (deg.)** | **-24.7** |
| **Flight time (yrs)** | **7.1** |
| **Total chemical ΔV (m/s)** | **200.0** |
| cruise (incl. nav) | 50.0 |
| Mnvrg @ asteroids | 150.0 |
| | |
| **LAUNCH VEH. CAPABILITY (KG)** | **1218.8** |
| LV margin (%) | 10.0 |
| LV margin (kg) | 110.8 |
| **ALLOWED INJECTED MASS (KG)** | **1108.0** |
| **Total propellant** | **355.5** |
| SEP | 285.0 |
| Chemical | 70.5 |
| **ALLOWED DRY S/C MASS (KG)** | **752.5** |

*Fig. X: 2003 7-year Vesta Sample Return*

## Spacecraft

Three-axis stabilization is preferred over spin stabilization for this mission, and both SEP and chemical propulsion systems are required, for the same reasons cited in the discussion of the multiple mainbelt asteroid rendezvous mission. The table in Fig. X shows that assuming 200 m/s chemical $\Delta V$ is required, the maximum allowed dry mass of the spacecraft at launch is 752.5 kg on the 2003 trajectory.

A mass summary is presented in Table X. Keeping a 10% launch vehicle margin allows a 19% mass growth contingency, which seems adequate in light of the overly conservative 10% launch vehicle margin. The only new technology assumed here is associated with development of sample acquisition and return hardware.

No effort was made to find the SEP system power level which maximizes mass performance. Some performance improvement is likely to result from examining power levels other than 6.75kw.

Attitude is determined to 0.01 deg. by a star tracker except during burn sequences when it is determined by a fiber-optic IMU. The orbiter carries one IMU; the lander carries another which serves as a backup. Two sun sensors are included. The C&DH system uses stacked MCM's. The main processor is a 33-MIP PowerPC 604.

| | Mass (kg) | % of dry mass (no cont.) | % of launch mass (no cont.) |
|---|---|---|---|
| Instruments | 34.0 | 5.4% | 3.1% |
| Lander | 118.0 | 18.7% | 10.7% |
| Sample return capsule | 30.0 | 4.8% | 2.7% |
| **Payload total** | **182.0** | **28.9%** | **16.4%** |
| | | | |
| Attitude Determination & Control | 16.0 | 2.5% | 1.4% |
| Command & Data Handling | 6.7 | 1.1% | 0.6% |
| Power | 108.9 | 17.3% | 9.8% |
| Propulsion-Chemical | 25.8 | 4.1% | 2.3% |
| Propulsion-SEP | 111.0 | 17.6% | 10.0% |
| Structure | 126.5 | 20.1% | 11.4% |
| Cabling | 23.8 | 8.0% | 2.1% |
| Telecom | 21.6 | 3.4% | 2.0% |
| Thermal | 7.4 | 1.2% | 0.7% |
| **Bus Total** | **447.8** | **71.1%** | **40.4%** |
| | | | |
| **Spacecraft Total (Dry)** | **629.8** | **100.0%** | **56.9%** |
| Contingency | 122.3 | **19.4%** | 11.0% |
| **Spacecraft with Contingency** | **752.1** | **119.4%** | **67.9%** |
| Chem. Propellant & Pressurant | 70.5 | 11.2% | 6.4% |
| SEP Propellant | 285.0 | 45.3% | 25.7% |
| **Launch Mass** | **1107.7** | **175.9%** | **100.0%** |
| | | | |
| **Launch Vehicle Capability** | **1218.8** | **193.5%** | **110.0%** |
| **Launch Vehicle Margin** | **111.1** | **17.6%** | **10.0%** |

*Table X: Spacecraft mass summary, SEP Vesta sample return mission*

Solar panels provide 6.75kw of power (end of life at 1AU) at an assumed 100 watts/kg. A rechargeable Li-ion battery (4.7 kg) is carried to provide power during launch. Two thermal batteries provide redundancy and can give brief periods of high power.

The SEP system consists of three gimbaled 30-cm. DS1 thrusters, two power processing units, one digital interface control unit, a propellant tank, and feed system components. The trajectory was computed assuming either 1 or 2 thrusters are operating during thrust periods; therefore, the spacecraft must carry at least two thrusters. To complete this mission, 285 kg of Xenon is required, which equates to a throughput of 95 kg of Xenon per thruster if 3 thrusters are carried. This slightly exceeds the 83 kg to which these thrusters were qualified before DS1 launch. However, a ground lifetime test is presently in progress to qualify those thrusters to 120kg Xenon throughput. The chemical propulsion system is a monopropellant hydrazine blowdown system with six 0.9-N thrusters on each of two lines (12 thrusters total). All components are off the shelf.

Structural mass (including cabling) was estimated to be 18% of non-structural injected ("wet") mass. Cabling was estimated at 8% of non-structural dry mass. Masses of the telecom subsystem, thermal protection subsystem, and sample return capsule are similar to estimates made for previous studies.

Lander

The lander's estimated mass of 118 kg includes 30% mass growth contingency. Changes in science requirements related to sample collection (e.g. sample size, number of samples, depth required) could have a great deal of influence on the lander's mass.

To aid in navigation during descent, hazard avoidance at landing, and rendezvous and docking with the orbiting spacecraft, the lander has imaging capability and terrain-mapping capability using a scanning lidar now under development for the Mars Sample Return Project. A star tracker and IMU determine attitude. Power is supplied by a non-rechargeable battery whose energy is 1600Wh at 28V, 57 A-h. This is sufficient to power the lander during its 27-hour mission (11 hours on the surface). A UHF transceiver and antenna relay data to Earth via the orbiter. Power from the orbiter heats the lander to 40 deg. C before its descent, allowing it to remain at an acceptable temperature on the surface for up to 24 hr. Propulsion is provided by a monopropellant hydrazine blowdown system.

Cost

Cost estimates from previous Team X studies and comparisons of factors affecting cost between this study and those previous studies indicate that the cost of this mission should fall near the top of the range of costs associated with Discovery missions.

Eliminating the X-ray/gamma-ray observations produces savings in both operations costs (since the duration of the mission at the asteroid can be reduced from ~1 year to a few months) and the cost of the instruments themselves.

## CONCLUSIONS

A SEP mission to rendezvous with both Vesta and Ceres appears feasible with current technology. Conducting global observations from orbit is preferred over landing a surface package to investigate a small surface area in-situ. Duration of the orbital reconnaisance mission is 1.5-2 years longer than the surface package mission because it measuring the elemental composition of the entire asteroid from orbit takes longer than measuring a small area on the surface. The increased operations costs of the orbital-only mission must be traded against the cost of building a surface package. No detailed assessment was made of the costs associated with this trade, however at a rough order of magnitude level these costs appear similar.

A SEP Vesta sample return mission appears feasible as well, but requires significant technology development in the area of sample acquisition and return. Further discussion on science requirements for sample retrieval (sample size, depth, number of sites) is also necessary to help determine the most effective technology development path.

Technology areas which could enhance both missions include development of solar panels, advanced SEP components with longer lifetimes, MCMs for use in the ADCS and C&DH subsystems, and reentry vehicle technology.

## ACKNOWLEDGEMENTS

Carl Sauer generated the trajectories discussed in this paper with the SEPTOP software which represents the product of his thirty years of experience in low-thrust trajectory design at JPL. The Advanced Projects Design Team ("Team X") at JPL also made significant contributions to this work.

## REFERENCES

1977 Bender paper

"Multiple Asteroid Rendezvous", a study of SEP mission options done by JPL's Advanced Projects Design Team ("Team X") led by T.H. Sweetser for R. Gershman, dated 11/25/96.

"Asteroid Rendezvous New Technology", a Team X study led by R. Bennett for A. Wolf, dated 5/9/97. This study reevaluated the 2-asteroid SEP mission discussed in the 11/96 study with new technology.

"ASAP Applications for SSE", a Team X study led by R. Oberto for R. Gershman, dated 6/29-30/98. This study examined Ariane piggyback options from geostationary transfer orbit (GTO). Mainbelt asteroid flybys using chemical propulsion and near-Earth asteroid rendezvous using solar sail were considered. Brief mention was also made of near-Earth asteroid rendezvous using SEP.

"Mission/System Design of AMBASSADOR, a Main Belt Asteroid Sample Return Mission", a Team X study done by JPL's Advanced Projects Design Team ("Team X") and led by R. Oberto, dated 8/19-22/97. JPL participated in this joint study with science and engineering students from the University of Arizona and Brown University. A SEP sample return mission to the asteroid Iris was studied.

Brian Wilcox subsurface explorer reference?

Personal communication with Al Metzger re: GRS data accumulation times?

Ultrasonic drilling reference, Y. Bar-Cohen?

Deep drilling reference, B. Dolgin?

Comet lander study for penetrator reference.